

6. Materials for Spacecraft

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6.1 Introduction

The general knowledge in this chapter is intended for a broad variety of spacecraft: manned or unmanned, low Earth to geosynchronous orbit, cis-lunar, lunar, planetary, or deep space exploration. Materials for launch vehicles are covered in chapter 7. Materials used in the fabrication of spacecraft hardware should be selected by considering the operational requirements for the particular application and the design engineering properties of the candidate materials. The information provided in this chapter is not intended to replace an in-depth materials study but rather to make the spacecraft designer aware of the challenges for various types of materials and some lessons learned from more than 50 years of spaceflight.

This chapter discusses the damaging effects of the space environment on various materials and what has been successfully used in the past or what may be used for a more robust design. The material categories covered are structural, thermal control for on-orbit and re-entry, shielding against radiation and meteoroid/space debris impact, optics, solar arrays, lubricants, seals, and adhesives. Spacecraft components not directly exposed to space must still meet certain requirements, particularly for manned spacecraft where toxicity and flammability are concerns.

Requirements such as fracture control and contamination control are examined, with additional suggestions for manufacturability. It is important to remember that the actual hardware must be tested to understand the real, “as-built” performance, as it could vary from the design intent. Early material trades can overestimate benefits and underestimate costs. An example of this was using graphite/epoxy composite in the International Space Station science racks to save weight. By the time the requirements for vibration isolation, Space Shuttle frequencies, and experiment operations were included, the weight savings had evaporated. [1]

6.2 Space Environment

Hardware exposed to space must withstand all aspects of the space environment. This includes vacuum, thermal cycling, charged particle radiation, ultraviolet radiation, and in some environments, plasma effects and atomic oxygen. Micrometeoroids and space debris particles may impact at high velocities; shielding is discussed in section 6.9. All of these may have significant effects on material properties either alone or in synergism.

The hard vacuum of space with its pressures below 10^{-4} Pa (10^{-6} Torr) causes some materials to outgas, which in turn affects any spacecraft component with a line-of-sight to the emitting material. This is discussed in more detail in section 6.4.

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Thermal cycling occurs as the spacecraft moves through sunlight and shadow while in orbit or conducts a “rotisserie” maneuver to keep solar exposure even. Thermal cycling temperatures are dependent on the spacecraft component thermo-optical properties, i.e. solar absorptance (α_s), or how much solar energy the material absorbs, and infrared emittance (ϵ_{IR}), or how much thermal energy can be emitted to space. The lower the ratio of α to ϵ , the cooler the temperature of the spacecraft surface. Thermal cycling can cause cracking, crazing, delamination, and other mechanical problems, particularly in assemblies where there is mismatch in the coefficient of thermal expansion.

Charged particle radiation includes protons and electrons with a wide range of energies. Spacecraft operating in or outside the Van Allen belts are exposed to much greater radiation levels than those in low Earth orbit. Charged particle radiation, along with ultraviolet radiation can cause cross-linking (hardening) and chain scission (weakening) of polymers, darkening and color center formation in windows and optics, and single event upsets in electronics. Radiation shielding is discussed in more detail in section 6.8.

Plasma refers to the ionized molecules in the upper atmosphere that have been excited by interaction with ultraviolet radiation and are affected by the Earth’s magnetic field lines. The plasma environment varies with altitude, latitude, time of day, and solar activity. Interaction with plasma and charged particles in the space environment contributes to the build-up of surface charge, especially in higher voltage systems. This surface charge can damage electronics, produce single-event upsets (SEU), trigger arcs in solar arrays or power systems, and cause dielectric breakdown of structure of surface coatings. NASA-HDBK-4006, *Low Earth Orbit Spacecraft Charging Design Handbook*, covers plasma interactions and mitigation techniques for high-voltage space power systems (>55 volts).

Atomic oxygen (AO) is produced when ultraviolet radiation reacts with molecular oxygen in the upper atmosphere. Currently only found in low Earth orbit between 100 and 1,000 km altitude, AO oxidizes metals, especially silver and osmium. AO reacts strongly with any material containing carbon, nitrogen, sulfur, and hydrogen bonds of 5 eV bond energy or less, meaning that most polymers react and erode away. Polymers containing fluorine, such as Teflon, react synergistically, where the reactivity to AO increases with longer exposure to ultraviolet radiation. [2] Some materials, such as ceramic coatings, can be bleached by exposure to AO.

6.3 Spacecraft Design and Materials Requirements

An old joke about the aerospace field is that when your paperwork weighs as much as your rocket, you are ready to launch. Good recordkeeping is absolutely vital to mission success. Not only a parts list but complete part identification and traceability – the test reports, inspection records, application, and location of an individual part or material are essential. Materials used in critical applications, such as life-limited materials, safety- and fracture-critical parts, liquid oxygen/gaseous oxygen (LOX /GOX) batch-sensitive materials, or materials requiring treatment to prevent hydrogen embrittlement, should be traceable by lot through all critical processing steps and the end-item application.

Vendors should understand the requirements for certifying fracture-critical hardware. In one notable case, a NASA contractor was convicted and fined heavily for improperly heat-treating, aging, and falsifying quality testing on aerospace hardware that was used in the Space Shuttle and Space Station programs, as well as commercial and military aircraft, and missile programs over a period of sixteen years. [3]

Fracture control requires that all space vehicle parts must be assessed to determine if structural failure in a space vehicle system would result in catastrophic failure. If the assessment determines that failure of the part would result a catastrophic failure, then that part must be subjected to full fracture control, including nondestructive evaluation (NDE). NDE methods for flaw or crack detection include eddy current, fluorescent penetrant, magnetic particle, radiography, and ultrasonics. Components that are exempt from fracture control are those that are clearly nonstructural and not susceptible to failure as a result of crack propagation, e.g., insulation blankets, electrical wire bundles, and elastomeric seals.

Designs should be developed using concurrent engineering practices, including manufacturability. Manufacturability is extremely important in spacecraft component designs to provide the best benefit to cost and schedule. Manufacturability factors to take into account are listed in Table 6.1.

Table 6.1 Factors to Consider for Manufacturability

Factor	Consideration
Drawings	Use geometric dimensioning and tolerancing Avoid double dimensioning Choose dimensions close to standard stock Use common angles if possible, 45° as opposed to 40°. Limit decimal places to that required Create a separate drawing for finishing if complex masking or several processes on one part.
Tolerances	Use realistic tolerance levels Be aware of tolerance stackup Consider inspection and tool access to areas for verification
Tapped Holes	Only design holes tapped 1.5x (or less) the diameter For blind holes, consider thread relief or do not tap to the bottom of the hole to prevent burr build up.
Internal Radii	Specify largest radii possible Use same radii when practical
Edges/Thickness	Reduce sharp corners or points, which are breakable Avoid thin webs and walls, deep holes to minimize distortion
Part Holding	Provide extra stock on all sides for clamping or chucking the work piece
Assembly	Design for disassembly Provide clearances for wrenches Include access holes where necessary
Materials	Select materials that are readily available to manufacturing Be aware that some materials are not available domestically, and their certifications can be difficult to get or are unreliable. Select materials with shortest processing time, machining, heat treating, <i>etc.</i>

	Select materials with the simplest storage requirements
Processes	Select processes that have been verified and are available to manufacturing
Composite Materials	Ensure selection of material system for which manufacturing has experience and verified processes
Surface Finishes	Specify minimum finishes
Coatings	<p>Use processes available and established in manufacturing</p> <p>Involve coating specialist, manufacturing and engineering before deciding on the best practice.</p> <p>Consider effects of coating processes on part dimensions, optical properties, <i>etc.</i></p> <p>Consider problems with coating holes, blind holes and difficult masking requirements.</p> <p>Use coating specific drawings if complicated masking.</p>
Heat Treat	<p>Consider using precipitation hardening alloys such as 17-4PH, 15-5MO, 12-8MO which only require a relatively low temperature of 480–620 °C (900–1150 °F) soak from one to four hours with an air cool in place of the common alloys like 4340 or 4130 steels, which require an austenitizing soak at 815-843 °C (1500-1550 °F) with a quick quench into oil followed by a tempering soak 480–600 °C (900-1100 °F). There is significant oxidation and scaling with the latter type of heat treat.</p> <p>On aluminum weldments, increase weld size or add gusseting rather than use heat treat to get the weldment back to a T6 condition, particularly with close tolerances or surface finish. This requires a solution treat at near melting temperature followed by a quick quench in water or other approved quenchant, followed by aging. Warping of the part and rupturing of enclosed areas can occur.</p>
Welding	<p>Utilize American Welding Society Standard Welding Procedure Specifications where possible.</p> <p>Keep weld length at a minimum</p> <p>Select a joint with least amount of filler</p> <p>Do not over weld</p> <p>Use square vs. round tubing for structural applications</p> <p>Design for inspectability and accessibility</p> <p>Allow for shrinkage and distortion.</p> <p>When using structural I and H beams be aware of the irregular dimensions of the mill supplied beams and call out your tolerances accordingly. It's hard to meet a +/- .030" tolerance when the beams may differ .250" from center line out to end of flange.</p>
Painting	<p>Ensure documented, verified processes available</p> <p>Provide proper level of surface cleanliness</p> <p>Consider the ability to get a paint gun perpendicular to the area being painted.</p>
Shop Capability	<p>Size and weight of component</p> <p>Forklift/crane limits</p> <p>Documented/verified processes</p> <p>Welding processes</p> <p>Sheet metal capability</p> <p>Surface treatment capability</p> <p>Heat treat sizes</p> <p>Cleaning/Painting size limitations</p>
Electrical/Electronic Components	Consider manufacturing capability and lead time requirements.
Packaging and Storage Requirements	<p>Size of component</p> <p>Environmental controls required</p> <p>Available space and equipment to meet storage requirements</p>

	Packaging material requirements, particularly for electrostatic sensitive parts
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As much manufacturing development as possible must be performed at full scale, including demonstration articles and mockups. When materials and process are scaled-up from small laboratory specimens to full-scale spacecraft components, a multitude of unforeseen problems can arise. Material properties can vary, machining and forming issues can arise, and when welding, the increased heat-sink of large components can cause weld property variations that often require modifications to tooling and/or weld process parameters. [4]

Shelf life of components must be considered during manufacturing. Organic-based materials have a limited shelf life and also limited static age life, i.e. time in a non-operating mode in an ambient environment. The properties of polymeric resins, catalysts, some lubricants, thin polymer films, sealants, adhesives, elastomers, and other materials may change slowly with time, even if sealed. Most manufacturers specify the shelf life, but often storage conditions are not specified. Generally, lower storage temperature extends the shelf life, as does minimizing exposure to light (both sunlight and fluorescent lighting). Another factor which influences the shelf life is the length of time that the product has been exposed to air and the constituents in it such as oxygen, moisture and other active agents.

6.4 Flammability, toxicity, and offgassing considerations

All spacecraft and ground support equipment materials must meet the criteria of NASA-STD-6001 (formerly NHB 8060.1), *Flammability, Offgassing, and Compatibility Requirements and Test Procedures*, for flammability, odor, offgassing, and fluid compatibility, depending on the environment to which the materials are exposed. Applicable environments are habitable environments, LOX and GOX systems, breathing gases, and reactive fluids. Table 6.2 is from NASA-STD-6001 and lists each test required depending on the environment for use. NASA maintains the Materials and Processes Technical Information System (MAPTIS) online database of materials test results.

NASA-STD-6001 Test 1 is upward flame propagation, or basic flammability. The control of flammability hazards on space hardware is extremely important because of crew safety. Two major lessons learned from the Apollo 1 fire investigations in 1967 were that ignition sources can never be eliminated and that propagation paths must be eliminated. If major propagation paths are eliminated, then, should a fire develop, it would be small and localized and would self-extinguish with little harm to the crew or vehicle systems. Limiting the total amount of combustible material is also important. Guidelines to control flammability can be found in NSTS 22648, *Flammability Configuration Analysis for Spacecraft Applications*, and MSFC-PROC-1301, *Guidelines for Implementation of Required Material Control Procedures*. Material behavior in oxygen-rich environments is discussed in Chapter 3.9 of this book.

Table 6.2 NASA-STD-6001 Required and Supplemental Tests for Each Material Use

Environment	Test Number	Type ¹	Title
Habitable Flight Compartments	1	R	Upward Flame Propagation
	2	R ² /S	Heat and Visible Smoke Release Rates
	3	S	Flash Point of Liquids
	4	R	Electrical Wire Insulation Flammability
	6	R ³	Odor Assessment
	7	R ³	Determination of Offgassed Products
	8	S	Flammability Test for Materials in Vented or Sealed Containers
	10	S	Simulated Panel or Major Assembly Flammability
	12	S	Total Spacecraft Offgassing
Other Areas ⁵	18	R	Arc-Tracking
	1	R	Upward Flame Propagation
	2	R ² /S	Heat and Visible Smoke Release Rates
	3	S	Flash Point of Liquids
	4	R	Electrical Wire Insulation Flammability
	8	S	Flammability Test for Materials in Vented or Sealed Containers
	18	R	Arc-Tracking
LOX and GOX Environments	6	R ³	Odor Assessment
	7	R ³	Determination of Offgassed Products
	13A	R	Mechanical Impact For Materials in Ambient Pressure LOX
	13B	R	Mechanical Impact for Materials in Variable Pressure LOX and GOX
	14	S	Pressurized Gaseous Oxygen Pneumatic Impact for Nonmetals
	17	R ⁴	Upward Flammability of Materials in GOX
Breathing Gases	1	R	Upward Flame Propagation
	6	R	Odor Assessment
	7	R	Determination of Offgassed Products
	13A	R	Mechanical Impact for Materials in Ambient Pressure LOX
	13B	R	Mechanical Impact for Materials in Variable Pressure LOX and GOX
Reactive Fluids	15	R	Reactivity of Materials in Aerospace Fluids

1 R -- Required Test; S-- Supplement Test

2 Required Test only for surface areas greater than 4 ft² (0.37 m² per each use.

3 Not required for materials inside hermetically sealed containers (see Section 2.1.3).

4 Not required for materials if they pass the criteria of Test 1 in that environment.

5 Includes all areas outside the habitable flight compartment.

NASA-STD-6001 Test 4 is electrical wire insulation flammability. An electrical fire during STS-61-A Spacelab D-1 mission in 1985 led to a better awareness of insulation fraying and abrading. During this incident when the insulation was worn through, the wire short-circuited and caught fire, but the breaker did not trip. Electrical breakers need to be readily accessible to cut the power to equipment on fire. The lessons learned from the Russian oxygen generator fire onboard the *Mir* space station in 1997 led to the addition of a containment shield and stricter quality control for oxygen canisters on the International Space Station. [5]

NASA-STD-6001 Test 7 and 12 determine offgassing. Offgassing or toxicity testing of materials to minimize trace contaminant gases insures air quality in habitable areas. Nonmetallic materials including coatings, adhesives, potting compounds, etc. may contain irritants such as formaldehyde, *n*-butanol, and aliphatic aldehydes. While activated carbon filters are effective at

removing some of these contaminant gases, they should not be relied on for long-duration manned missions. Bakeout of offgassing material is recommended to drive off the volatiles, usually for 48 hours at 50 °C (122 °F).

Outgassing still refers to the volatiles released from a material but is a separate test from offgassing for materials used in a vacuum environment. All external spacecraft materials must be tested at a minimum, ASTM-E-595, *Standard Test Method for Total Mass Loss and Collected Volatile Condensable Materials from Outgassing in a Vacuum Environment*, with not more than 1.0 % total mass loss (TML) and 0.1% collected volatile condensable material (CVCM) recommended. Depending on the spacecraft contamination control plan, ASTM-E-1559, *Standard Test Method for Contamination Outgassing Characteristics of Spacecraft Materials*, testing may be required. Rather than just TML and CVCM, ASTM-E-1559 provides outgassing rates over time and at different temperatures for better dynamic modeling of any contaminants.

Materials in line-of-sight to sensitive optics may be tested per MSFC-SPEC-1443, *Outgassing Test for Non-Metallic Materials Associated with Sensitive Optical Surfaces in a Space Environment*. This test was developed during the Hubble Space Telescope program to ensure that materials that passed ASTM-E-595 did not evolve enough volatiles to impact optical performance in the ultraviolet wavelengths. Reflectance measurements are made on a magnesium fluoride/aluminum mirror before and after exposure in vacuum to the candidate material, with a change of more than 3% being cause for rejection of the material for the proposed application.

Outgassing, or molecular contamination can severely impact spacecraft performance. For example, the navigation camera on the Stardust spacecraft was blurred by contaminant deposition and required several heating cycles to improve image quality. [6] The heating cycles did not remove all of the contamination, so there was limited resolution in the images Stardust took of the Wild 2 comet nucleus during flyby.

One of the lessons learned from the Shuttle-Mir program that benefited the International Space Station was the impact of contamination on radiator coatings and solar arrays. Samples of Z-93 white ceramic thermal control coating flown on the Passive Optical Sample Assembly (POSA) – I experiment were degraded to nearly end-of-life solar absorptance after only 18 months in space. Electron Spectroscopy for Chemical Analysis (ESCA) and ellipsometry analysis of nearby optical samples indicated 5000 – 10,000 Å of contamination had been deposited on one side of the experiment. These samples were line-of-sight to a stowed solar array with large amounts of unbaked silicone. [7]

The ISS program took steps to minimize molecular contamination by both materials selection and thermal vacuum bakeout, with a requirement of less than 130 Å contamination per year. Samples of Z-93, AZ93, and other white ceramic thermal control coatings on the Materials on International Space Station Experiment (MISSE) indicate the success of the contamination control plan for ISS. Localized contamination can still be found on ISS (fig. 6.1), but overall, the performance of the radiators and solar arrays has not been impacted by molecular contamination at time of publication.

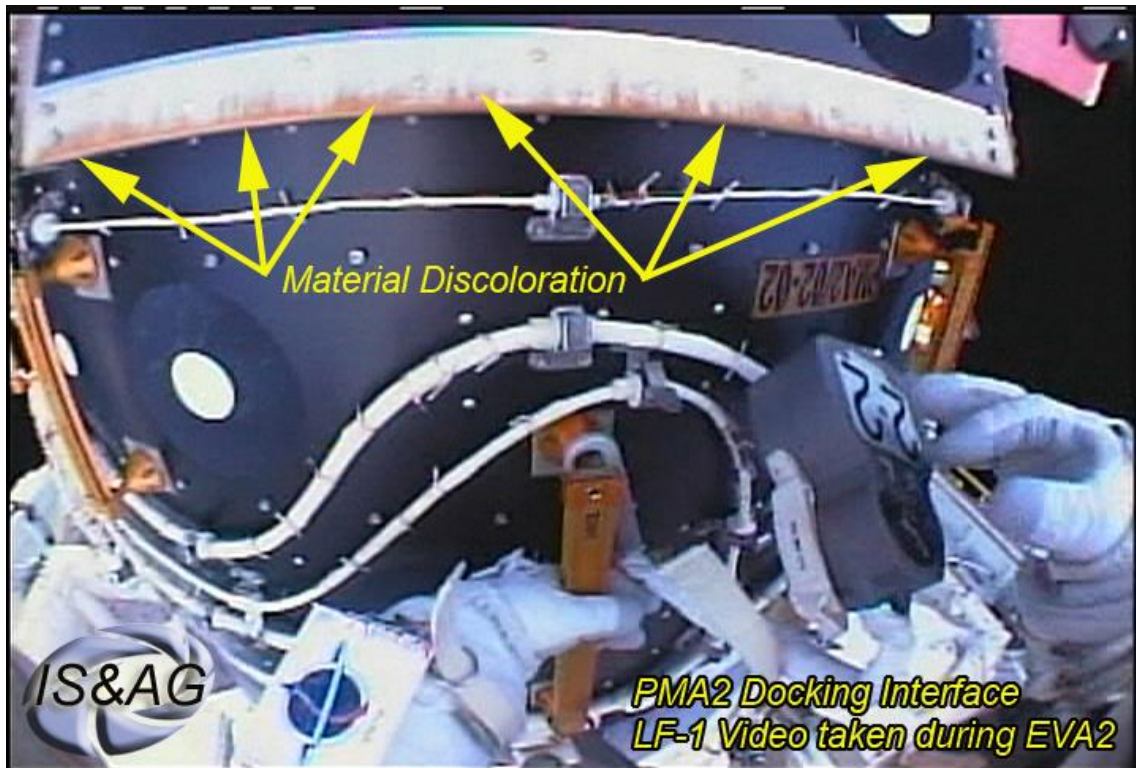


Figure 6.1 Localized contamination on ISS

Outgassing may be detrimental to materials even in the internal spacecraft environment. Silicone outgassing hampered the performance of beds used for carbon dioxide removal in the Environmental Control and Life Support System (ECLSS) on ISS. These beds were vented through the station's vacuum system, promoting the outgassing. Thermal vacuum bake of the beds prior to flight significantly improved the performance.

6.5 Structural materials (also refer to Chapter 3)

Strength-to-weight ratio is usually the critical factor in choosing structural materials for spacecraft. Static and dynamic loads must be considered, along with thermal performance, corrosion protection, manufacturability, reparability, and cost. Values for allowable properties of structural materials in their design environments should be taken from *Metallic Materials Properties Development and Standardization* (MMPDS, formerly MIL-HDBK-5), MIL-HDBK-17, *Plastics for Flight Vehicles*, and MIL-HDBK-23, *Structural Sandwich Composites* or other approved sources. Use-dependent properties must also be considered, such as dielectric constant for radomes or gas permeability for tanks.

High strength alloys of aluminum, titanium, and stainless steel have been in common use for decades. However, the 5000-series aluminum alloys containing more than 3% magnesium shall not be used in applications where the temperature exceeds 66 °C (150 °F), because grain boundary precipitation above this temperature can create stress-corrosion sensitivity or exfoliation. This includes 5083-H32, 5083-H38, 5086-H34, 5086-H38, 5456-H32, and 5456-

H38. For the same reason, the 300 series corrosion-resistant (CRES) stainless steel should not be used at temperatures above 370 °C (700 °F) for extended periods of time.

With their higher chromium and nickel content, austenitic stainless steels are more resistant to stress corrosion cracking than ferritic and duplex stainless steels. In general, titanium alloys and high nickel content alloys are resistant to stress corrosion cracking. MSFC-STD-3029, *Guidelines for the Selection of Metallic Materials for Stress Corrosion Cracking Resistance in Sodium Chloride Environments*, states, “Many copper alloys containing more than 20 percent zinc are susceptible to stress corrosion cracking, even in the presence of alloying additions which would normally impart resistance to stress corrosion.” The standard also warns that protective coatings may only delay onset of stress corrosion, not entirely prevent it. Surface treatments such as carburizing or nitriding may increase susceptibility to stress corrosion cracking.

Aluminum-lithium alloys have 10% or more weight savings over standard aerospace aluminum alloys. For example, aluminum-lithium alloy was used in the Superlightweight Tank (SLWT) for the Space Shuttle, with a weight savings of 7,000 lbs. over the original External Tank. Friction stir welding has been used on aluminum-lithium alloy as well as other aluminum alloys previously considered unweldable. Friction stir welding requires neither inert shielding gas nor filler material, reduces the number of weld defects, and has higher weld joint strength over fusion joining processes where the metal is melted.

Space structures that demand tight tolerances in coefficient of thermal expansion, such as telescope optical benches, are usually made of composite materials. A wide variety of fibers, including graphite, boron, fiberglass, aramids, and carbon are available, as are many polymer resin systems, including epoxy, phenolic, polyimide, and polysulfone. The fiber may be in tow, tape, sheet, or woven form for traditional polymer-matrix composites. Metal-matrix composites (MMCs) and ceramic-matrix composites (CMCs) may have particulate or fiber reinforcement, where the fiber can be continuous or discontinuous (chopped fibers or whiskers). These are used where high toughness is needed.

For non-metallic materials, atomic oxygen erosion may be a concern if used in low Earth orbit, as is chain-scission or cross-linking of polymer chains in ultraviolet and particle radiation environments. These are usually surface effects, but a long exposure to atomic oxygen may compromise the strength of thin composites. For example, composites on the leading edge of the Long Duration Exposure Facility (LDEF) satellite lost a full ply to atomic oxygen over 5.8 years. [8] A high radiation environment may result in strength loss and embrittlement for some polymeric materials.

Honeycomb structures have also been used for their excellent stiffness. These can be with either composite or metallic facesheets and cores. A lesson learned from the X-33 honeycomb composite tank is the need for closed cell cores (dependent on application) and redundant permeation barriers in cryogenic applications to eliminate cryopumping. Cryopumping is “the influx of gas into an unclosed volume resulting from the vacuum generated when cryogenic temperatures liquefy and condense the gas on the cryogenic boundaries of that volume.” [9] When the X-33 tank was filled during testing, liquid hydrogen leaked through microcracks in the inner facesheet, and the nitrogen purge gas was pulled through microcracks in the outer facesheet

to fill the core. When the tank was drained and started warming, the core pressure increased to the point of failure, when the outer facesheet and core separated and peeled away from the inner facesheet. (fig. 6.2) (Also see chapter 7)

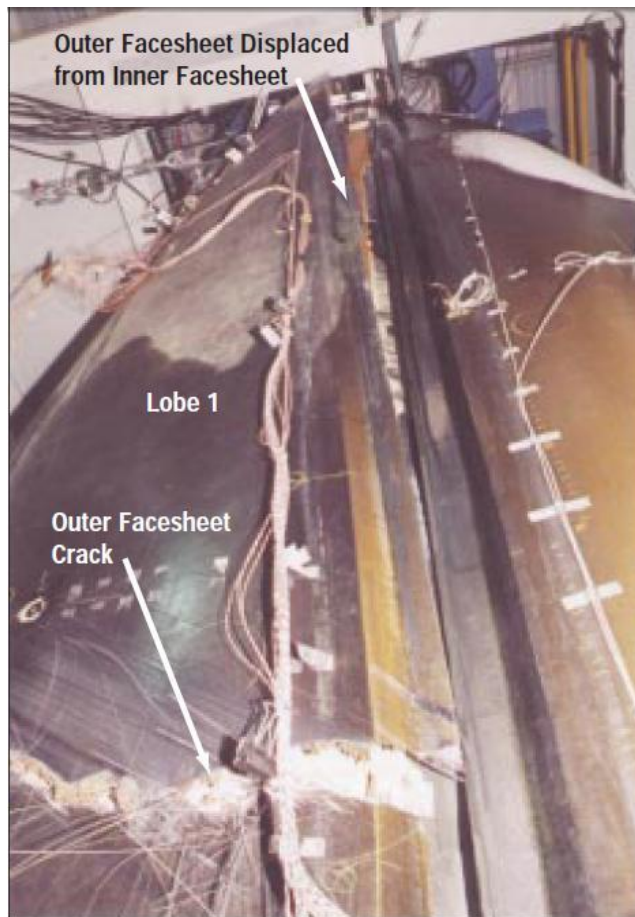


Figure 6.2 X-33 cryogenic tank after failure.

6.6 Thermal control materials

Passive thermal control of a spacecraft can be by a surface treatment, application of a coating, or covering with a multi-layer insulation blanket. Surface treatment of metals is generally required to prevent corrosion prior to launch. Environmental protection directives have led to increasing restrictions on hexavalent chromate use, which is commonly used in chemical conversion coatings. Newer conversion coatings may contain trivalent chromium or no chromium at all. Corrosion and space environmental effects testing of these coatings are continuing. While chemical conversion coatings per MIL-C-5541 provide more than adequate corrosion protection, they are usually very low in thermal emittance and may not meet temperature requirements on orbit. Manned spacecraft that may require extravehicular activities (EVAs, also known as spacewalks) will have more stringent thermal control requirements for touch temperatures, generally -118 to +113 °C (-180 to +235 °F).

Better passive thermal control in terms of absorptance and emittance may be achieved with anodizing per MIL-A-8625. MIL-A-8625 calls out three types of anodize – Type I chromic acid,

Type II sulfuric acid, and Type III hard anodize. Reduction of chromate use has virtually eliminated Type I chromic acid anodize. Type II sulfuric acid anodize that will be exposed to space should be processed with a hot water seal, not nickel acetate, as the residue from the nickel acetate seal yellows after a brief exposure to ultraviolet radiation. Type III hard anodize is also performed with sulfuric acid but usually at lower temperatures to get a thicker oxide layer, when needed for wear resistance. However, hard anodize should be carefully evaluated before use on parts subjected to fatigue. Boric/sulfuric acid and phosphoric acid anodize have been tested in space and may be useful when the desired thermal properties cannot be met by sulfuric acid anodized aluminum (generally $\alpha = 0.45/\epsilon = 0.80$) without compromising corrosion protection.

When a lower α/ϵ ratio is needed, passive thermal control coatings or paints are used. Those qualified for use on spacecraft generally have one of four types of binder – polyurethane, epoxy, silicone, or potassium silicate. Acrylic-based paints have very poor performance in space and should not be considered. Polyurethane and epoxy-based coatings have limited life in low Earth orbit, because atomic oxygen will erode the binder, leaving only pigment particles behind. [10] Silicone, usually a low-outgassing type, is used when some UV darkening can be allowed or if the geometry is complicated enough where coating adhesion is a concern. Coatings with silicone binders should be used with caution when in line-of-sight to sensitive optics. Potassium silicate coatings can be difficult to apply and are sensitive to contamination, but they have the advantage of proven durability in space for long periods of time and are generally considered safest around optics. [11] Manufacturer-recommended cure times for coatings should be followed, e.g. putting Z-93 potassium silicate coating in vacuum before the 7-day cure is complete will cause the coating to crack and debond. Manufacturers' cure conditions may not be adequate to insure the coating does not become a contamination source to sensitive optical surfaces. Materials may require thermal vacuum bakeout and should be tested under conditions equivalent to on-orbit extremes.

Passive thermal control coatings and anodizes may be electrically insulative enough to build up surface charge in the space environment. Sufficiently high surface charge can even break down the anodized layer on aluminum (fig. 6.3) or result in operational anomalies. NASA RP-1390 details a number of satellites that had system failures due to the plasma environment. [12] Some of these spacecraft charging events caused loss of control of the satellite. These events sometimes required station-keeping fuel to stabilize, therefore shortening the life of the satellite. Static-dissipative materials or conductive coatings can bleed off this surface charge before damage occurs. Indium tin oxide-coated thin films have been used for many years, but special care must be taken not to crack the coating. Also, fiberglass cloth with conductive thread has been used.

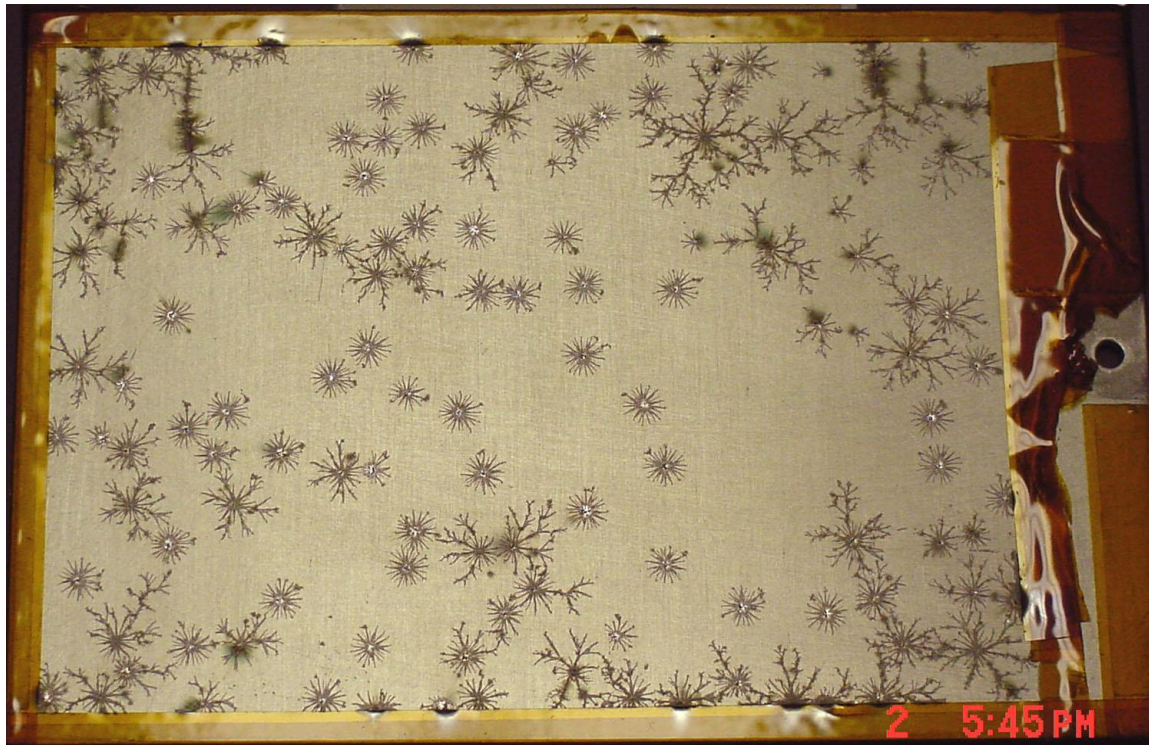


Figure 6.3 Breakdown of anodized aluminum in plasma environment

Conductive or static-dissipative thermal control coatings usually have conductivity on the order of 10^5 to 10^8 ohms/square. Coatings loaded with carbon for conductivity (e.g. Electrodag 501) should not be used in low Earth orbit, as the carbon will be eroded by atomic oxygen, but they may be acceptable for other environments. It is of utmost importance that the coating demonstrate conductivity in high vacuum after adequately conditioning (at least 24 hours as a minimum). In the past, some coatings were advertised as conductive when they were actually dependent on water content for their conductivity. As the water was removed by vacuum, these coatings became insulative.

Multilayer insulation (MLI) blankets use multiple reflectors, usually thin polymer films with vapor-deposited metal on one or both sides, separated by lightweight, low-thermal conductivity materials. These thin films are fragile, as evidenced by the damage by purge on the Huygens probe (fig. 6.4), so a durable outer cover is usually added to the MLI blanket, and an inner cover may be added as well. Common MLI blanket materials and their properties can be found in NASA TP-1999-209263, "Multilayer Insulation Material Guidelines." It should be noted that MLI blankets composed of Mylar reflector layers should be kept below 120°C (250°F) because of shrinkage and loss of tensile strength.



Figure 6.4 A worker repairs the multi-layer insulation on the Huygens probe after an abnormally high pressure purge ripped the delicate material.

MLI blankets require atmospheric pressure of less than 10^{-5} torr to prevent convection and gas conduction between the reflector layers. The reflector layers may have perforations or porolations to allow better venting. MLI blankets should not be pulled taut; pinch points, heat shorts, and cut-outs should be minimized. Organic threads should not be used in low Earth orbit unless they are adequately protected from atomic oxygen erosion. Hook and loop tape fastener (e.g., Velcro or Aplix) seams should have some blanket overhang, preferably 1", to prevent atomic oxygen erosion of the polymeric hooks and loops down to stubs. It is recommended that unmated hook and loop tape fasteners not be exposed to more than 1×10^{20} atoms/cm² of atomic oxygen.

Electrical grounding required for MLI depends on the size of the blanket and the environment. Proper grounding requires that the spacer netting be cut away and grounding tape, such as aluminum tape with conductive adhesive, applied in a continuous accordion between the

reflector layers of the blanket. (fig. 6.5) A hole is punched through all of the layers, and a grounding bolt with washers, eyelet terminal, and locking nut is installed. A single conductor wire is generally used between the eyelet terminal and ground.

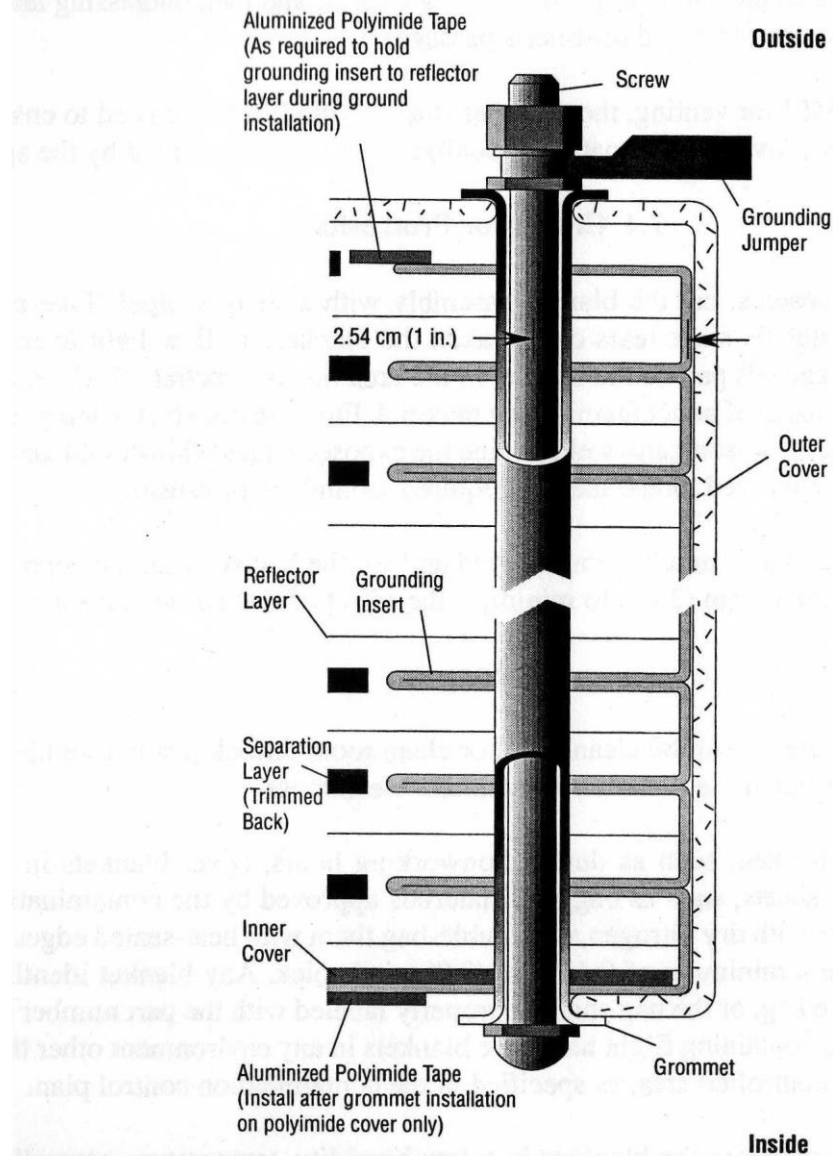


Figure 2.5 Electrical grounding design for multilayer insulation blanket. [13]

In addition to MLI blankets, the large dewar on Gravity Probe-B used vapor-cooled metallic shields. The dewar contained superfluid helium cooled to near absolute zero. As the helium was slowly vented during spacecraft operations, it was circulated through the shields. This design was successful in that the helium onboard stayed liquid for 16 months.

6.7 Thermal protection materials

Thermal protection materials are separate from thermal control materials in that thermal control materials are used to moderate on-orbit temperatures, and thermal protection materials are generally for higher temperatures, such as around engine exhaust or for reentry. These temperatures may reach 2,800 °C (5,070 °F). Heatshields may be made from reusable materials, such as tiles or ceramic-matrix composites, or one-time use materials, such as ablatives. Heatshield material selection is dependent on the peak heat flux and stagnation pressure during reentry (fig. 6.6), as well as mechanical strength, density, entry angle, and the shape of the heatshield (i.e. blunt-body, sphere-cone, biconic, or non-axisymmetric).

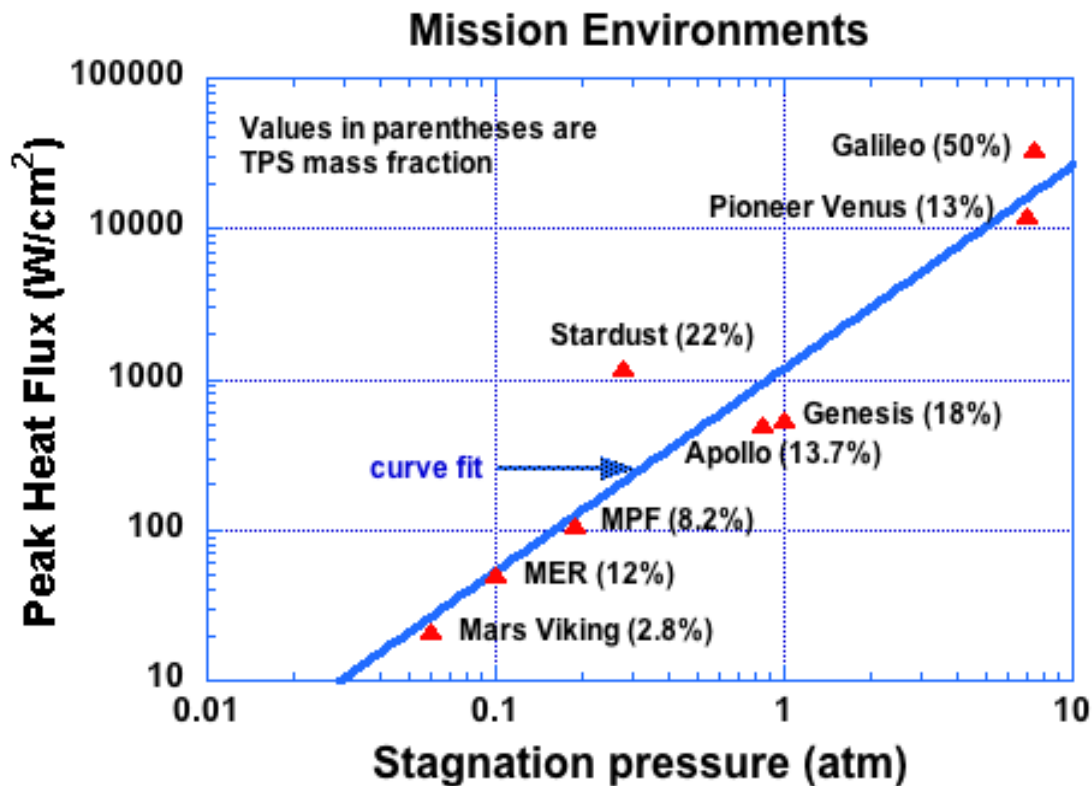


Figure 6.6 Peak heat vs. stagnation pressure for various missions. MER is Mars Exploration Rover; MPF is Mars Pathfinder. (Figure courtesy of Bernard Laub/NASA Ames Research Center)

Determining the thickness of the heatshield must balance weight versus the uncertainty of performance. For example, the ablation modeling of the Galileo entry predicted that the nose of the shield would ablate considerably more than the shoulder region. However, the data from the ablation sensors revealed that this was not the case. The TPS of Galileo recessed about 44 mm in the shoulder region, compared to the predicted ablation distance of about 33 mm, leaving a margin of only 10 mm. [14, 15] The stagnation point recession model predicted that the nose would recede about 88 mm, but the measured recession value was 41 mm.

Silica ceramic tiles, or high-temperature reusable surface insulation (HRSI), were developed for use on the Space Shuttle. They are lightweight, can be formed in various densities, and can handle reentry temperatures up to 1,260 °C (2,300 °F), but they are fragile and easily damaged.

They also require a coating to be waterproof. Toughened unipiece fibrous insulation (TUF) tiles are stronger and tougher.

The X-37B (fig. 6.7) Orbital Test Vehicle uses similar heatshielding as the Space Shuttle, with lightweight tiles on the belly and flexible insulation blankets for lower temperature areas. Quilted blankets of woven silica fiber, silica batting, and aluminoborosilicate fiber were used on the Space Shuttle where reentry temperature stayed below 649 °C (1,200 °F).



Figure 6.7 X-37B after landing at Vandenberg AFB

Reinforced carbon-carbon (RCC) was used in the nose cap and wing leading edges of the Space Shuttle. It can be used where reentry temperature exceeds 1,260 °C (2,300 °F). Carbon fibers in a silicon carbide matrix (C/SiC) were used in the nose cap, leading edges and steering flaps for the X-38 vehicle, though this was only ground-tested and not flown. Multilayer high temperature ceramics such as zirconium boride and silicon carbide [16, 17] or nanocomposites [18] can be used to protect carbon/carbon composites against oxidation.

Ablative heatshields are usually a honeycomb material with a resin or polymer injected into each cell. Materials include the Avcoat used on the Apollo capsules, phenolic-impregnated carbon ablator (PICA) utilized by the Stardust sample return capsule, and SLA-561V used on the Viking landers.

6.8 Radiation shielding

Space radiation can be from galactic cosmic rays, trapped radiation, auroral radiation (polar orbits only), and solar flare. Most of the radiation is in the form of protons, electrons, and alpha

particles (helium ions), but heavier ions do impact spacecraft. In addition, the reaction of radiation with higher Z (higher atomic number) materials creates Bremsstrahlung radiation. Low Z materials, such as polyethylene, are good protection from beta particles (electrons) and Bremsstrahlung radiation. A composite of low and high Z materials, particularly where the high Z material is sandwiched between two layers of low Z materials can improve shielding per areal weight performance for shielding electronics.

Adequate radiation shielding for long-term manned missions still needs to be developed. Astronauts may take cover in a “storm cellar” with extra radiation shielding during a solar flare. Liquid hydrogen and other fuel tanks as well as water tanks have been suggested for multi-purpose radiation shielding of astronauts, as has incorporating lunar or Martian regolith into a polymer binder for in-situ resource utilization. [19] Research is continuing in this field, particularly with hydrogenated graphite nanofiber and other nanocomposites. Active shielding generated by either magnetic or electrostatic fields has also been proposed but may be impractical due to interference with avionics and communications.

6.9 Meteoroid/ Orbital Debris Shielding

Hypervelocity impacts by meteoroids and orbital debris are a significant hazard for spacecraft. Average velocity for orbital debris is 10 km/sec, while meteoroids can travel as fast as 60 km/sec. For comparison, a high-speed rifle fires at ~1 km/sec. The Russian satellite Kosmos-1275 was at an altitude known to be populated with space debris (977 km) and in a high inclination orbit that increased relative velocities to that debris. In July 1981, the satellite broke up into over 200 trackable fragments, likely due to collision with space debris. While large pieces of orbital debris, usually 10 cm and up, can be tracked and avoided, smaller particles are still capable of damaging hardware, particularly windows. Over the 30-year life of the Space Shuttle, at least 45 windows had to be replaced because of impacts. (fig. 6.8) Windows on the International Space Station are triple-pane and, in the case of the Cupola (fig. 6.9), have shutters to minimize likelihood of impacts.

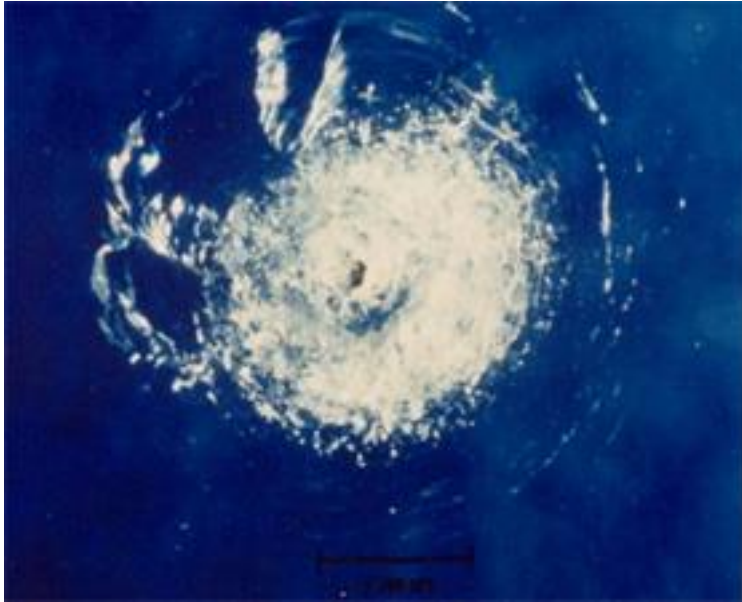


Figure 6.8 Space debris impact on Shuttle window

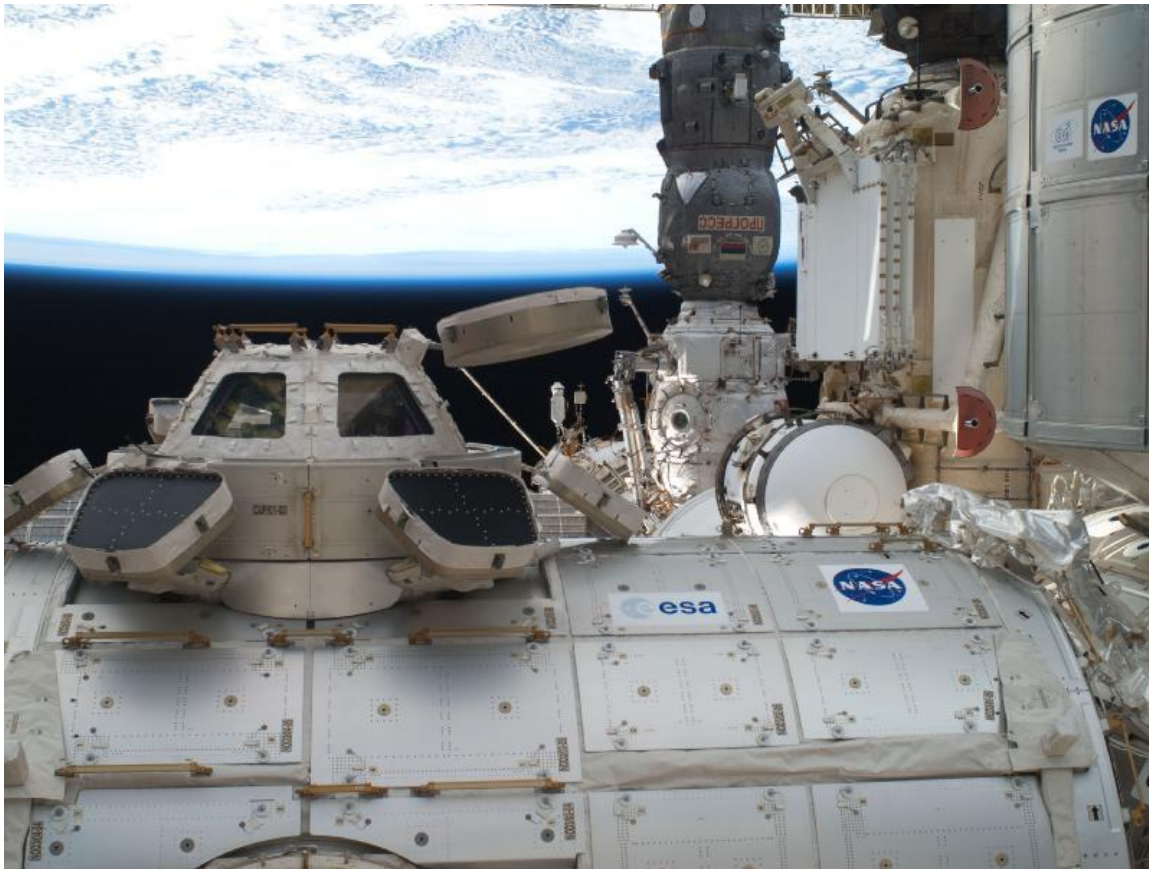


Figure 6.9 ISS Cupola with open shutters (Photo courtesy of NASA)

The fundamental method of protecting a spacecraft from hypervelocity impact is the Whipple shield. The sacrificial bumper breaks up the impacting particle into smaller pieces or, in case of

extremely high impact velocities, vapor. The number of bumpers and the spacing between them and the pressure wall or other critical hardware will be determined by desired probability of no penetration (PNP) and spacecraft weight and volume constraints. The most common Whipple shield material is aluminum. However, in the case of the Deep Impact mission, the impactor had five sheets of copper to differentiate it from any comet material. Various composites and honeycomb materials have also been used for bumpers.

Thermal protection and hypervelocity impact protection can be combined in a multi-layer insulation blanket. The International Space Station (ISS) utilizes a “stuffed Whipple” design, which includes a blanket of Kevlar and Nextel between the bumper and the pressure wall. [20] Other aramid materials such as Nomex, Twaron, and Aracon (which is coated Kevlar) may also be used. It should be noted that an MLI blanket may provide better protection when spaced evenly between the bumper and the pressure wall.

For manned spacecraft, the Whipple shield design will also depend on what material is being used for the pressure wall. A study of the aluminum alloys 2219-T87 and 5456-H116 indicated increased petaling (where the aluminum peels back like flower petals, which can then detach and cause damage) and spallation in the 5456-H116 dual-wall at impact velocities higher than 7 km/sec. [21] Recommended reading on this topic is the Handbook for Designing MMOD Protection. [22]

6.10 Optical materials

Window materials, such as quartz, fused silica, and sapphire, should have low impurity content. Space radiation reacts with the impurities to form color centers, degrading the transmission. Coating the window with cerium oxide will also improve the durability in space.

Magnesium fluoride has been flown in both window form and as a layer over aluminum for a mirror. Workmanship is important, as some magnesium fluoride samples have endured the space environment quite well, while the Hubble Space Telescope’s original Wide Field/Planetary Camera flood mirror suffered blistering and peeling. Magnesium fluoride does react with atomic oxygen to form an oxide layer, as does lithium fluoride, calcium fluoride, and barium fluoride. Zinc selenide, which is used in some infrared optic applications, also reacts with atomic oxygen to form a thin oxide layer. This oxide layer increases absorptance.

Most telescope designs do not directly expose the mirror(s) to the space environment, but there is space environmental effects data for gold, platinum, iridium, and nickel optical coatings. [23] Nickel reacts with atomic oxygen to form a thin oxide layer, with an accompanying drop in reflectance. Osmium should not be used in low Earth orbit applications as it reacts strongly with atomic oxygen and erodes away.

An optical solar reflector (OSR) is a second-surface mirror where the metal, usually aluminum or silver, is protected by a layer of quartz. OSRs usually have low solar absorptance and high infrared emittance and can be used for thermal control in a high radiation environment where metalized Teflon is contraindicated due to the increase in emittance with exposure.

6.11 Solar array materials

The International Space Station uses silicon solar cells with ceria-doped borosilicate coverglass for power generation. Performance monitoring over the years indicates a 1.5 to 3.5% power loss/year in a fairly low radiation environment.[24] The Solar and Heliospheric Observatory (SOHO) is orbiting the L1 Lagrangian point, with a much higher radiation environment, but about the same degradation per year of 2%. [25] SOHO's solar arrays are back surface reflector silicon solar cells and ceria-doped microsheet (CMX) cover glass. Degradation due to proton radiation was observed, particularly during solar eruptions in July 2000 and November 2001. These solar eruptions with accompanying protons cause displacement damage, where radiation interacts with the solar cell lattice producing defects which reduce the solar cell's output voltage and current.

One of the solar arrays flown on the *Mir* space station for over 10 years was returned to Earth and was found to have 58% degradation, or 5.8% per year.[26] This is the same low Earth orbital environment as the International Space Station. A combination of factors led to this higher rate of degradation, including electrical arcing, meteoroid/debris impacts, degradation of interconnects, and heavy silicone contamination of the solar array.

Although the behavior of silicon solar cells is well-known, silicon has been replaced with higher efficiency multi-junction solar cells on recent spacecraft, including the Mars Exploration Rovers *Spirit* and *Opportunity*. Multi-junction solar cells typically have layers of germanium, gallium arsenide, and gallium indium phosphide. The Deep Space 1 probe included the Solar Concentrator Array with Refractive Linear Element Technology (SCARLET) solar arrays with silicone linear Fresnel lenses for light concentration deployed over GaInP₂/GaAs/Ge solar cells (fig. 6.10).[27]

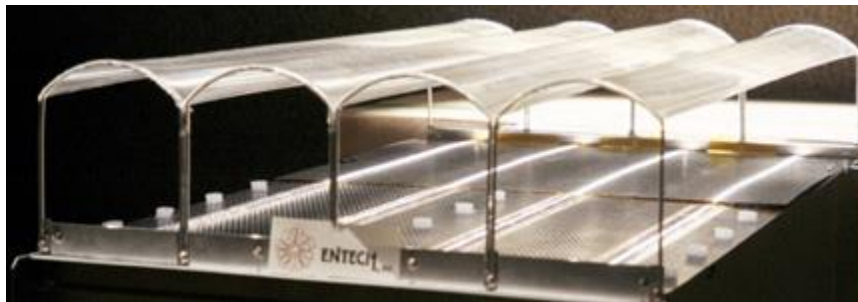


Figure 6.10 SCARLET solar array with curved silicone Fresnel lenses above multi-junction solar cells

Multi-junction solar cells have the same challenges in a radiation environment as silicon solar cells, with displacement damage. Data from Spectrolab, a leading manufacturer of solar cells, indicates approximately 1% per year performance loss in space for their high-efficiency triple-junction GaInP/GaAs/Ge solar cells over 15 years, more than likely with ceria-doped borosilicate cover glass to increase the durability.[28]

6.12 Lubricants

The vacuum of space presents a challenge for keeping mechanisms lubricated. Hermetically sealing moving parts to keep the lubrication intact may be possible in some cases but is usually considered impractical. When choosing a liquid lubricant (oil, fluid, or grease), the lower the volatility, the better. Also, the component design must include provisions to retard creep, evaporation, and draining by gravity. The temperature extremes of space must be considered also, as liquid lubricants have a limited temperature range of operation. Some commonly used liquid lubricants include perfluoropolyethers (PFPE's), multiple alkylated cyclopentane (MAC), and polyalphaolefins (PAO's).

A better choice for moving mechanical assemblies exposed to the space environment is a solid lubricant. Solid lubricants include molybdenum disulfide, tungsten disulfide, niobium diselenide, graphite powder, silver, Teflon (polytetrafluoroethylene), and nylon. Unbonded solid lubricants include loose powder, brush-on, or spray-on lubricants. Burnishing, or applying loose powder onto a component with pressure, is another application method but is subject to wide variability in thickness and wear performance. This type of lubricant is not recommended for space-exposed applications. Bonded solid lubricants may be in a resin binder or inorganic binder. McMurtrey [29] notes that the wear behavior of resin-bonded lubricant films is different from other solid lubricants because of initial wear-in, loss of loose material on the bearing surface, and compaction. Inorganic-bonded solid lubricants in general do not perform as well as resin-bonded at room temperature but may be more suitable for high temperatures. Sputtered molybdenum disulfide may be difficult to apply for certain parts but has excellent coefficient of friction. Graphite-based lubricants will not work in space due to the lack of moisture and will become abrasive. The molybdenum disulfide lubricants are the better choice for space use especially in sliding applications. [30]

Lubricants containing chlorofluorocarbon constituents should not be used with aluminum or magnesium if high shear stresses can be imposed. Temperature limits should also be considered, such as for graphite which loses its absorbed water at higher temperatures and is not recommended for use above 1000 °F (538 °C). Molybdenum disulfide oxidizes at 750 °F (399 °C) and may also be affected by high humidity. Storage in an inert atmosphere is recommended for sensitive parts lubricated with molybdenum disulfide to decrease exposure to humidity.

The Solar Alpha Rotary Joint (SARJ) on the International Space Station is an example of insufficient understanding of tribological interaction, and therefore a weak lubrication design, which resulted in severe surface distress on orbit. The joint allows the solar arrays to rotate to maximize incident sunlight. It was noted during flight operations that more power was required to rotate the starboard SARJ than the port side and that there was more vibration. Inspection of the starboard SARJ during a spacewalk revealed a large quantity of metal particles (fig. 6.11) that was actually spalled nitride layer of the rolling surface where the trundle bearing assemblies contacted and rolled on the steel truss structure or race-ring joint.[31]

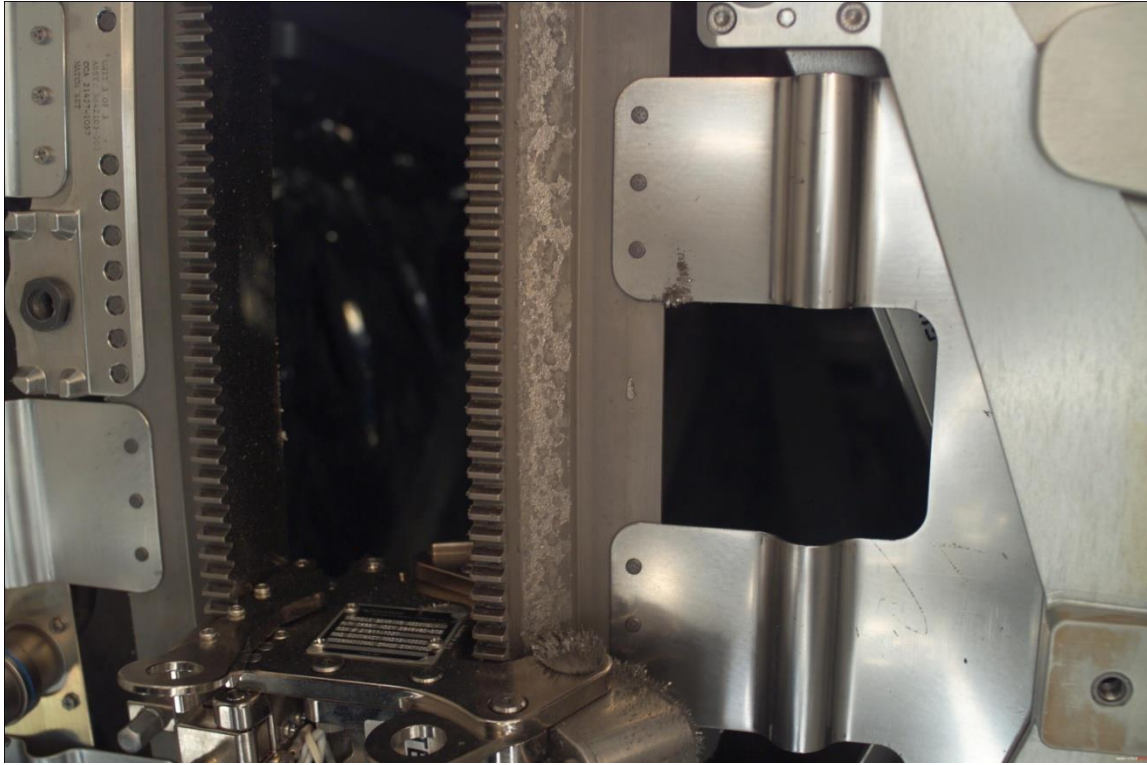


Figure 6.11 Damage in Solar Alpha Rotary Joint (SARJ) on International Space Station (ISS)

These trundle bearings that formed the structural connection between the large rotating and non-rotating truss sections also included tapered roller assemblies that allowed the “alpha” rotation of the arrays to track the sun. The trundle bearings, similar to track follower bearings that hold an amusement roller coaster cart to a track, were designed for ostensibly rolling contact for low friction. But the on-orbit operation included just enough detrimental slip that the solid film lubricant scheme, a thin gold coating on the rollers, could not reduce the total slip/roll interface friction to a tolerable level. This combined friction ultimately exceeded the capability of the hardened nitride rolling surface to withstand the subsurface stresses, resulting in a unique, massive spalling of the nitride layer. The damage required three more spacewalks to remove the trundle bearings, clean out the loose, hard metallic spalling debris, replace the trundle bearings, and lubricate the system, this time with vacuum compatible grease that also included a molybdenum disulfide additive. The starboard SARJ, as of this writing, is operating fine with the repair, although the rolling surface is much rougher and softer than the intended design with the nitride layer intact. While a rigorous ground test program was implemented to predict the life of the lubricant fix, ultimately time will tell if the repair was effective. Also, there are valid concerns over long term outgassing of openly greased surfaces of such magnitude that are in line-of-sight to contamination-sensitive surfaces on ISS. Meanwhile, the port side SARJ, which was also lubricated with the perfluorinated MoS₂ grease, continues to operate fine with no evidence of spalling damage.

This SARJ anomaly example does not promote the use of a specific grease lubricant over other solid films, greases, or their base oils. Rather the anomaly highlights the need to fully understand the system level tribological behavior and desired lubrication regime (boundary to

fully separating lubricant films) to properly design mechanisms that will perform with acceptable friction and tolerable wear. All too often a lubricant is called upon to correct weak tribological designs, much like treating a symptom versus the responsible disease. Such “lube fixes” are often not even considered a principal design change as there is no perceived change to the “hardware.” Incorporation of a better tribological awareness in all mechanisms in the concept and design phase can save a lot of major headaches later on. Even so, all boundary lubrication regime tribological contacts (solid film lubricants, starved fluid lubricant films, slow moving high load, etc.) or other questionable tribological couples should be tested at the highest system and fidelity level practical. Modeling or analytically predicting behavior of boundary lubricated contacts requires understanding the interaction of more variables than can usually be adequately known.

6.13 Seal materials

Seal materials discussed in this section are limited to those used to maintain vehicle pressurization, pneumatics, and hydraulics. These may not be suitable for applications related to propulsion systems (see chapters 10-13). Any seal material used in hydraulic systems should be checked for fluid compatibility. In general, seals are made of metal or elastomer. Metal seals can be made of soft aluminum, copper, or stainless steel, though nickel, Monel, and Inconel have been used in the past. A wide variety of elastomeric seal materials are available, including butyl rubber, silicone, Viton, Teflon, Kel-F fluoropolymer, and Neoprene. Seal materials to be used on windows should be thermal vacuum baked prior to flight to minimize outgassing.

6.14 Adhesives

Two classes of adhesives will be covered in this section – structural adhesives, such as those used in honeycomb laminate manufacture, and non-structural adhesives, such as the pressure-sensitive adhesive used for thermal control tapes. The coefficients of thermal expansion (CTE) of adhesives and substrates should be evaluated to ensure that a CTE mismatch does not lead to problems. When using an adhesive on two substrates with different CTE, the adhesive should have an adequate amount of elasticity (rubber or polyurethane) to compensate. Surface preparation requirements must be specified and rigorously enforced to maintain bond characteristics. Failures in large propulsion components have been attributed to inadequate control of the bonding process, usually either in the surface preparation steps or in the storage and handling of the adhesive material including mixing and application of the 2-part epoxies.

Structural adhesives can be two-part epoxies, modified phenolics, or thermosetting resins. Cytec FM-300 and 3M AF-191 film adhesives have been used for honeycomb core/face bonds, Hysol EA9394 paste for external splices. The epoxy family of thermosetting adhesives exhibits good solvent resistance and good elevated temperature properties to ~350 °F. Disadvantages are two-component mixing requirements, limited pot life, exothermic reactions, and deterioration of properties in hot and wet environments. Adhesives that utilize epoxy-amine or amide polymer systems will react with copper and result in blue to green corrosion products. Large propulsion components that used adhesively bonded components generally require long pot and working life and tailored cured properties. These materials many times are not available off the shelf,

therefore the adhesive manufacturers should be involved in identification or development of an adhesive that meets the design requirements.

Pressure-sensitive adhesives are usually either acrylic or silicone-based. 3M 966 (sometimes referred to as Y966) is commonly used in tapes and can be used on either metal or composite surfaces. 3M 9406PC is also acrylic and outgasses less than the 966. 3M 9703 is an electrically conductive acrylic adhesive. Arclad 8026, NuSil CV-1144, and DC93-500 are common silicone adhesives for spacecraft applications. Cleanliness and proper preparation of the surface prior to tape or adhesive application is essential.

Other adhesive materials include polyurethanes, cyanoacrylates, and polyimides. The polyurethane adhesive family has excellent low-temperature flexibility. Disadvantages are complex mixing and application procedures, short pot life, moisture sensitivity in cured and uncured states, and poor elevated temperature performance. The cyanoacrylate family has good strength, fast setting, ease of use, and exhibits good adhesion to metal substrates. Disadvantages are high cost, poor durability on some surfaces, and limited solvent and elevated temperature resistance. Polyimides have excellent high-temperature capability (greater than 370 °C/700 °F) with high strength. Disadvantages are high cost, low peel strength, poor processing characteristics, low tack, and elevated temperature cure of ~343 °C/650 °F.

6.15 Lessons Learned

Dr. Mac Louthan [32] teaches how catastrophic failures in materials engineering can be traced to six root causes that are applicable to spacecraft design, development, manufacture and assembly. Perhaps keeping these causes in mind can help to prevent future failures.

1. Deficiencies in Design.

The meteoroid shield and one of the solar panels on Skylab were damaged during launch because venting had not been adequately addressed. Sixty-three seconds after launch, internal pressure built up behind the meteoroid shield until the shield moved into the supersonic air stream. The shield was then torn from the Orbital Workshop (OWS) part of Skylab, damaging one solar panel and blocking a second solar panel from fully deploying. This left the space station with little power or thermal control. Two EVAs were required, the first to deploy a temporary sunshade, the second to free the blocked solar array

Venting analysis had been performed but assumed a sealed aft end on the meteoroid shield. The need to keep this seal was not adequately communicated to aerodynamics, structural design or manufacturing personnel. [33]

2. Improper Materials Selection

During the period May-July 1998, one or more of the redundant spacecraft control processors (SCPs) failed in each of three Hughes HS-601 commercial communications satellites. This resulted in one of the satellites being removed from service. Three SCP failures were attributed to intermittent or continuous short circuits caused by the growth of conductive filaments, known as "tin whiskers," from the tin-plated surface of an electronic assembly or its cover.[34]

Tin plating is used on many electronic devices and space hardware for corrosion protection and ease of soldering. Pure tin plating is subject to a crystal instability problem. The normal cubic crystal form of pure tin is unstable and has a tendency to change to tetragonal at temperatures below 10 °C (+50 °F). This instability increases with decreasing temperatures to a maximum at -40 °C (-40 °F). This crystal transformation results in an expansion of the crystal lattice. This expansion causes the tinplating to crack and spall off the substrate. This phenomenon is called "tin pest" or "tin disease.", It can be prevented by adding small amounts of lead, bismuth, or antimony to the tin (GSFC Materials Tip 021, Problems with Pure Tin Coatings).

Tin whiskers create a potential electrical problem whenever printed wiring boards with tin plating are used for extended duration, regardless of the environment and whether the boards are conformal coated. Tin whisker formation has been widespread, and numerous incidents of electrical equipment failures have been attributed to its formation; therefore, copper conductors and circuit paths on printed wiring boards should not be electroplated with pure tin. Potential tin whisker growth can be eliminated by incorporating a minimum of 1.5-percent lead into the electrodeposited tin and reflowing the tin alloy.

3. Defects in Material

In the 1980's, Martin Marietta received a lot of two-part primer for the External Tank in which the wrong solvent reducer had been shipped, even though the can was labeled with the correct material and the shipping paper showed that the correct material had been shipped. This primer was flight critical for bonding the urethane foam to the aluminum tank substrate. The error in labeling was not caught until after a technician sprayed a 1,000 sq ft tank dome, noted that the material did not behave normally and reported it. The dome area had to be cleaned by hand sanding and scrubbing with Scotch Brite and a solvent. [35] Not only does this example point to the need for experienced personnel during manufacture and inspection, but it also shows the need for a culture where problems can be identified and taken care of, rather than ignored, covered up or glossed over.

Many times material performance can be affected when manufacturers change materials formulations or manufacturing process steps without the knowledge of the material user. Numerous occurrences were noted during the Space Shuttle Program when unreported manufacturing changes resulted in unexpected materials performance during flight. What manufacturers or raw material suppliers may consider as insignificant changes can require expensive hardware rework or disposition rationale development when discovered downstream in the manufacturing or launch preparation flow.

4. Improper Processing

During the Apollo 13 mission, the number 2 oxygen tank exploded during a "cryo stir" procedure. The tank held a "slush" of liquid oxygen with a fill line and heater running down the center, and the cryo stir procedure turned on an internal fan to keep the slush from stratifying. The original specifications for the tank and heater assembly were for 28 V DC power. This was later changed to 65 V DC, however the thermostatic switches for the heater were missed in the changeover. These switches could accommodate the higher voltage during tank pressurization because they normally remained cool and closed, but they could not open without damage.

During assembly, the tank was accidentally dropped, loosening the fill tube assembly and allowing gas leakage. Because the tank could no longer drain properly, an improvised detanking procedure at KSC was developed. During detanking, as the switches started to open at the upper temperature limit, they were welded permanently closed by the too-high voltage. Failure of the switches led to the heater being on for much longer and much higher temperatures than expected, melting the fan motor wire insulation. The higher tank temperature was not noticed because the temperature gages only measured to 80 °F. In flight, when the crew performed the cryo stir, the wire short circuited, arced and ignited its insulation, triggering the explosion in the oxygen tank.[36]

5. Inappropriate Assembly

The first test of the Tethered Satellite System during STS-46 in 1992 was halted when the reel mechanism jammed. Only 260 meters (853 ft) of the 20 km (12.43 mi) tether was deployed. An investigation found a protruding bolt from a late-stage modification of the tether reel system.

6. Inadequate Service

The high-gain antenna for the Galileo spacecraft did not deploy properly after launch. The reason for this is not precisely known, but it is likely that the three years in storage while waiting for launch after the *Challenger* disaster led to the jammed mechanism. The most likely culprit is evaporation of the lubricant, but damage could have occurred while in storage or during one of the unplanned trips between the Jet Propulsion Laboratory in California and Kennedy Space Center in Florida during the launch delay. The backup low-gain antenna saved the science mission. Because of this incident, NASA communications satellites with similar antenna designs have the pins replaced just before launch, and new lubricant is added.[37]

6.16 Concluding Remarks

As with any engineering task, the requirements must be properly understood, and the materials must be selected to meet those requirements with appropriate safety margins. Design of a spacecraft should take into account the interaction of all the components – structures, thermal, propulsion, avionics, guidance, navigation, and control (GN&C), etc. The full life cycle of the design should be considered including safety, verification, reliability, operability, and maintainability. Decisions on materials and manufacturing should not be “stovepiped”.

Spacecraft materials should be selected based on the environment and the time of exposure, with either durability in that environment or known degradation for appropriate endurance and end-of-life properties. They should be correctly assembled and maintained, with proper compatibility with surrounding materials and corrosion prevention. Attention to detail is never wasted.

Good communication is essential. Communication with manufacturers and vendors can prevent problems with changes in processing or discontinued products. Interaction with other disciplines can help with understanding interfaces, identify deficiencies and improve requirements. Ongoing dialogue can keep a small change from becoming a big problem.

To confine our attention to terrestrial matters would be to limit the human spirit.

— Stephen Hawking

6.17 Acknowledgments

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